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RESEARCH MEMORANDUM

THE EFFECTS OF CIRCULAR END PLATES ON THE LIFT, DRAG,
AND PITCHING MOMENT AT SUBSONIC AND SUPERSONIC
SPEEDS ON A MODIFIED TRIANGULAR WING HAVING
AN ASPECT RATIO OF 2, A TAPER RATIO
OF 0.33, AND A 45° SWEPT
LEADING EDGE

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SUMMARY

An investigation has been made of the effects of circular end plates on the aerodynamic characteristics of a modified triangular wing of aspect ratio 2 and taper ratio 0.33.

The experimental results show that the addition of end plates produces so slight a difference in the total change in static longitudinal stability with Mach number in the range from $M = 0.6$ to $M = 1.9$ that no beneficial effects on the trim drag may be realized.

The addition of circular end plates influences other aerodynamic characteristics in the following ways:

1. Increase in lift-curve slope
2. Increase in minimum drag
3. Decrease in induced drag
4. Increase in maximum lift-drag ratio at Mach numbers less than 0.87, and a decrease for Mach numbers greater than 0.87
5. Increase in the lift coefficient at which maximum lift-drag ratios occur

INTRODUCTION

The use of end plates as a method of improving the aerodynamic characteristics of swept wings at low subsonic speeds has been investigated in references 1 and 2. These investigations showed that the end plates, which acted as a barrier to the spanwise flow around the outboard portion of the span, caused an increase in lift-curve slope and a rearward shift in the center of pressure.

The effect of end plates on the center-of-pressure position has suggested the possibility of using them as a means of overcoming large changes in the static stability that occur between subsonic and supersonic speeds. The increase in lift due to the addition of end plates is so distributed on a modified triangular wing that there is a rearward shift in the center of pressure. At subsonic Mach numbers the wing area influenced by the end plates was reasoned to be greater than the relatively small area in the tip Mach cones which would be affected at supersonic speeds; therefore, since the wing had a modified triangular plan form, it was reasoned that there would be a greater rearward movement of the center of pressure at subsonic speeds than at supersonic speeds.

Accordingly, an experimental investigation of the effects of circular end plates on the longitudinal stability and other aerodynamic characteristics of a modified triangular wing was undertaken in the Ames 6- by 6-foot supersonic wind tunnel. The results are reported herein.

SYMBOLS

| | |
|-----------------------------------|--|
| b | wing span, ft |
| \bar{c} | mean aerodynamic chord, $\frac{\int_0^{b/2} c^2 dy}{\int_0^{b/2} c dy}$, ft |
| c | local wing chord, ft |
| l | length of body including portion removed to accommodate sting, ft |
| $\left(\frac{L}{D}\right)$ | lift-drag ratio |
| $\left(\frac{L}{D}\right)_{\max}$ | maximum lift-drag ratio |

| | |
|------------------------|---|
| M | Mach number |
| q | free-stream dynamic pressure, lb/sq ft |
| R | Reynolds number based on mean aerodynamic chord |
| S | total wing area, sq ft |
| y | distance perpendicular to plane of symmetry, ft |
| α | angle of attack of body axis, deg |
| C_D | drag coefficient, $\frac{\text{drag}}{qS}$ |
| $C_{D_{\min}}$ | minimum drag coefficient |
| C_L | lift coefficient, $\frac{\text{lift}}{qS}$ |
| C_m | pitching-moment coefficient, $\frac{\text{pitching moment}}{qS\bar{c}}$ |
| $\frac{dC_L}{d\alpha}$ | lift-curve slope, per deg |
| $\frac{dC_m}{dC_L}$ | slope of curve of pitching-moment coefficient as a function of C_L (static-longitudinal-stability parameter) Pitching moments for the wing with end plates were taken about the 25-percent mean aerodynamic chord, while the pitching moments for the wing alone were taken about the 20-percent mean aerodynamic chord. |

MODEL AND APPARATUS

The experimental investigation was conducted in the Ames 6- by 6-foot supersonic wind tunnel. In this wind tunnel, the Mach number can be varied continuously and the stagnation pressure can be regulated to maintain a given test Reynolds number. Further information about this wind tunnel is presented in reference 3.

The model was mounted in the tunnel on a sting-type model support whose pitch plane was horizontal. The 4-inch-diameter, four-component balance, described in reference 4, was used to measure the aerodynamic forces and moments. Plan and front views of the model and certain model

dimensions are given in figure 1. The model used in the investigation was an aspect ratio 2, taper ratio 0.33, plane wing, with a 45° swept leading edge composed of 3-percent biconvex sections with a rounded nose modification. The profile of the rounded nose section forward of the midchord was elliptical, and the tangent to the airfoil section at the midchord was horizontal (see profile in ref. 5). The end plate was a flat circular plate, 7 inches in diameter, with a single beveled edge, and was symmetrically located on the wing tip chord.

The wing and end plates were constructed of solid steel. The body was constructed of steel and aluminum. The surfaces of the body, wing, and end plates were polished smooth.

TESTS AND PROCEDURE

Range of Test Variables

The lift, drag, and pitching moment were measured at angles of attack from -4° to 17° . The results were obtained for Mach number ranges of 0.61 to 0.93 and 1.2 to 1.9 for Reynolds numbers as listed in table I, and for $M = 0.91$ at a Reynolds number of 6.6×10^6 .

Reduction of Data

The test data have been reduced to standard NACA coefficient form. The factors which could affect the accuracy of the results, together with the corrections applied, are the same as those in a previous investigation of the basic wing (ref. 5). The same data corrections were applied for the wing alone and for the wing-end-plate combination, except that the moment axis position for the wing alone was placed at 20 percent of the mean aerodynamic chord instead of 25 percent as for the wing-end-plate combination. The pitching-moment transfer factors were adjusted accordingly. The selection of the two different moment axis positions provides the same static margin at $M = 0.6$ for both wings.

RESULTS AND DISCUSSION

The basic data plots of lift coefficient versus angle of attack, drag coefficient, and pitching-moment coefficient for the modified triangular wing with and without end plates are shown in figure 2 over the range of test Mach numbers. It was found that the nonlinearities in

the pitching-moment curves over the range of lift coefficients $-0.05 < C_L < +0.05$ for the wing at subsonic Mach numbers were reduced by increasing the Reynolds number as shown in figure 3. Increasing the Reynolds number had little influence outside this lift-coefficient range. Because of power limitations, most test data were taken at low Reynolds numbers and, therefore, the slope of the pitching-moment curves was taken by placing a line through the values of pitching-moment coefficient at ± 0.1 lift coefficient.

A comparison of the data for the model with and without end plates would be more realistic from a design viewpoint if both configurations were given the same static margin at $M = 0.6$. Both the moment axis positions were, therefore, adjusted to give a minimum static margin of approximately 5 percent (wing alone at 20-percent mean aerodynamic chord and wing-end-plate combination at 25-percent mean aerodynamic chord).

Figure 4 presents the comparison of the static margin for the wing with and without end plates over the Mach number ranges 0.61 to 0.93 and 1.2 to 1.9. The data indicate that there is no significant difference in the total change in the static margin in going from low subsonic to supersonic speeds. The end plates, therefore, do not act to reduce the pitching moment to be balanced by a control surface; therefore, if it is assumed that the effectiveness of a control surface is the same for both configurations at a trim condition, no beneficial effects are realized on drag.

Figure 4 also indicates that the addition of end plates increases slightly the degree of static longitudinal stability at high subsonic Mach numbers and at zero lift coefficient, but the basic data of figure 2 indicate that at high subsonic Mach numbers and at a lift coefficient of approximately 0.3 the static longitudinal stability is adversely affected. This break in the pitching-moment curve gives the model with end plates undesirable longitudinal stability characteristics at lift coefficients within the operating range of an airplane.

The effect of end plates on the lift-curve slope for subsonic speeds has been investigated both theoretically and experimentally in references 1 and 2. These references express this effect on the basic wing as an increase in effective aspect ratio, which increases the lift-curve slope. It was found in the present investigation (fig. 5) that the lift-curve slope was increased at all Mach numbers.

In the low lift coefficient range the drag coefficients are increased due to the drag of the end plates (fig. 5); but for the higher lift coefficients, the reduction in the induced drag coefficient is greater than the increase in the drag coefficient at low lift coefficients (fig. 2).

An increase in the minimum drag due to the addition of end plates is realized at all the Mach numbers (fig. 5). The drag penalty may not be as great, however, as indicated in figure 5, because the vertical end plates may be incorporated into an airplane design for directional stability.

The addition of end plates shows in figure 5 a small increase in the maximum lift-drag ratio for Mach numbers below 0.87 and a slight decrease for the Mach numbers tested that were above 0.87. The addition of end plates also increases the lift coefficients at which the maximum (L/D) ratio occurs for all the Mach numbers tested (fig. 5).

CONCLUSIONS

The addition of circular end plates to a modified triangular wing had the following effects:

1. There was no significant difference in the total change in the static margin in going from low subsonic to supersonic speeds. As a result, the moment to be trimmed out and, hence, the trim drag that would be produced by a trimming device were not favorably affected.
2. The static longitudinal stability at high subsonic Mach numbers and at a lift coefficient of approximately 0.3 was adversely affected.
3. The lift-curve slope at zero lift was greater for all Mach numbers tested.
4. The drag coefficient was increased at low lift coefficients but was decreased at high lift coefficients.
5. The maximum lift-drag ratio was caused to increase slightly for the Mach number range 0.61 to 0.87 and to decrease slightly for the Mach number ranges 0.87 to 0.93 and 1.2 to 1.9.
6. The lift coefficients at which the maximum (L/D) ratios occurred were increased.

Ames Aeronautical Laboratory
National Advisory Committee for Aeronautics
Moffett Field, Calif., Oct. 14, 1953

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TABLE I.- REYNOLDS NUMBERS AT VARIOUS TEST MACH NUMBERS

| Mach number | Reynolds number |
|-------------|-------------------|
| 0.61 | 4.8×10^6 |
| .71 | 4.8 |
| .81 | 4.8 |
| .91 | 4.8 |
| .93 | 4.8 |
| 1.2 | 3.8 |
| 1.3 | 3.8 |
| 1.4 | 3.8 |
| 1.5 | 3.8 |
| 1.6 | 3.1 |
| 1.7 | 3.1 |
| 1.9 | 3.1 |

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Equation of fuselage radii:

$$\frac{r}{r_0} = \left[1 - \left(1 - \frac{2x}{l} \right)^2 \right]^{3/4}$$

All dimensions
shown in inches.

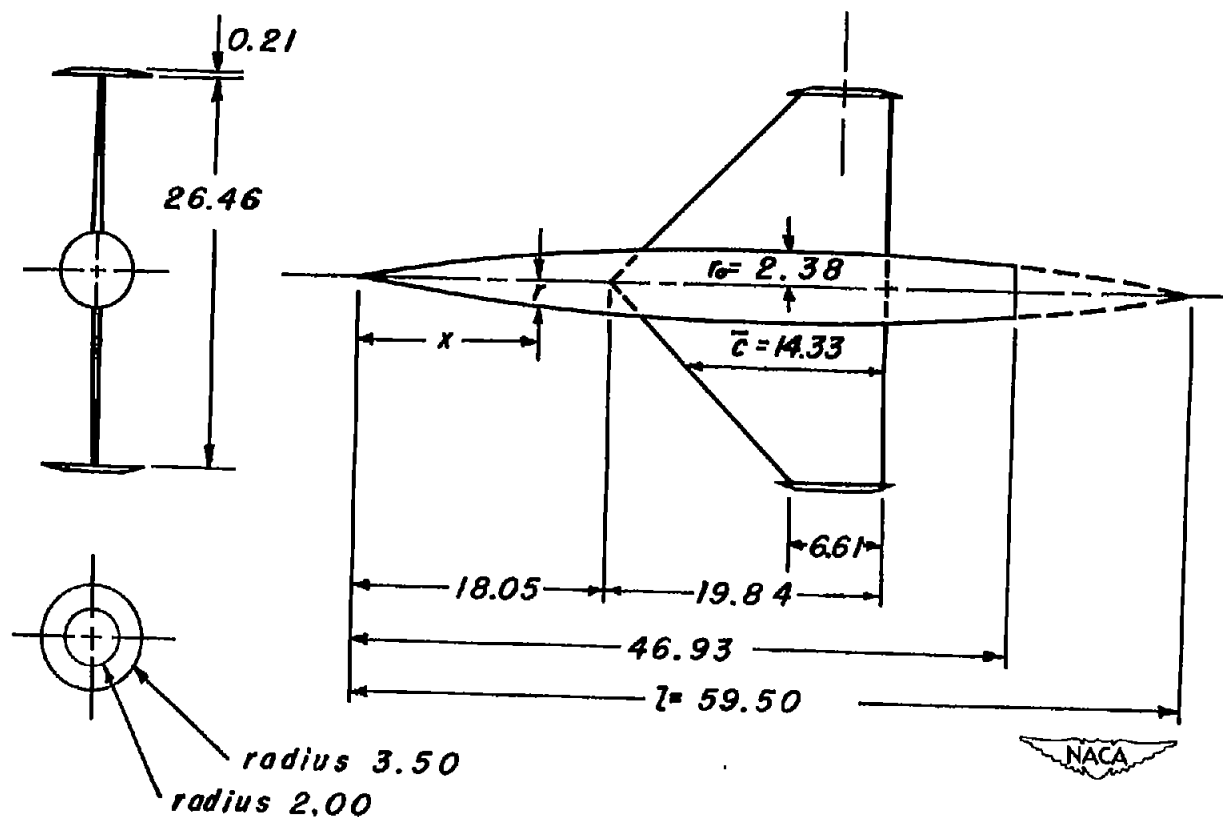


Figure 1.- Plan and front view of a modified triangular wing model.

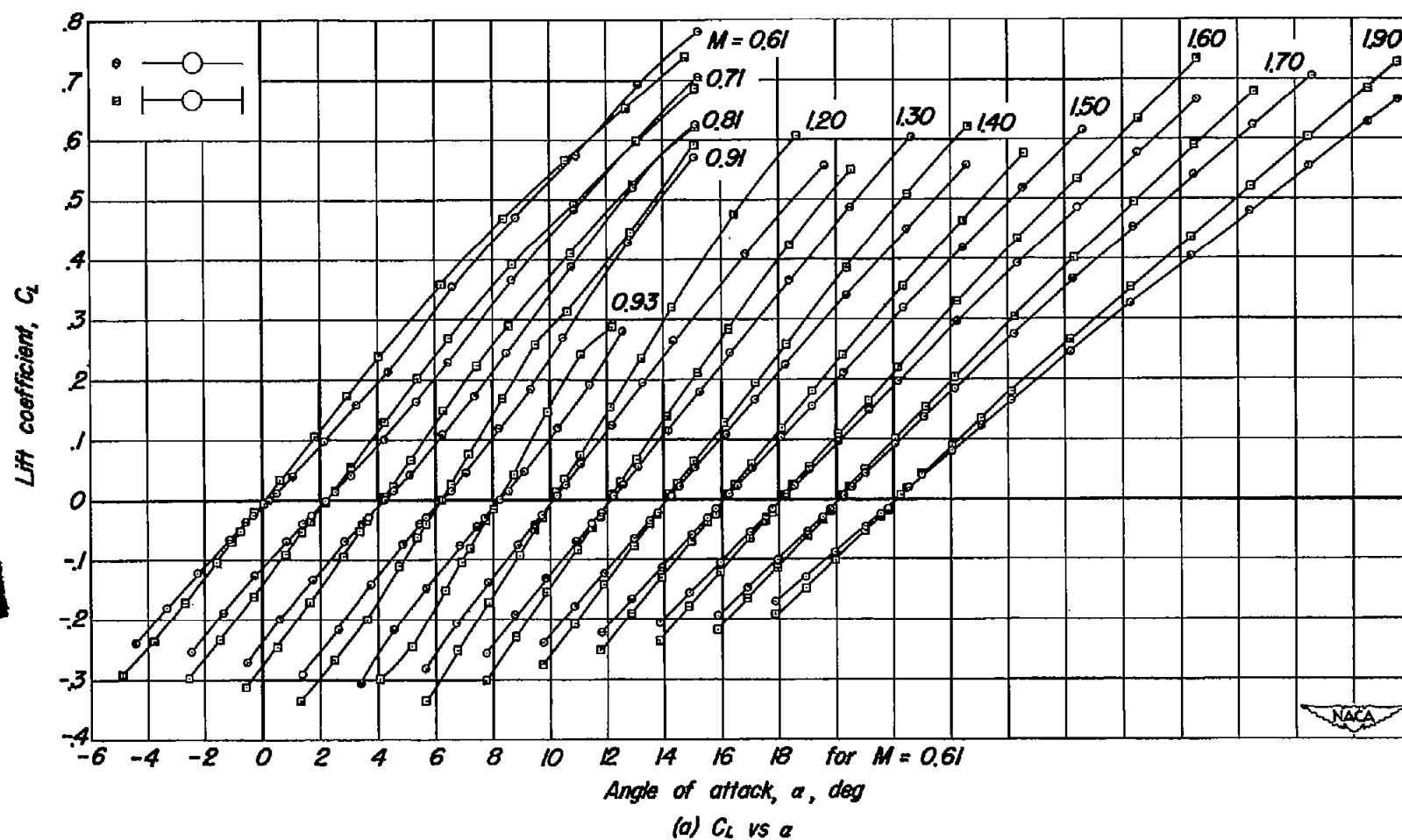


Figure 2.- The effect of the addition of end plates on the variation of the aerodynamic characteristics with lift coefficient at various Mach numbers. Reynolds numbers as listed in table 1.

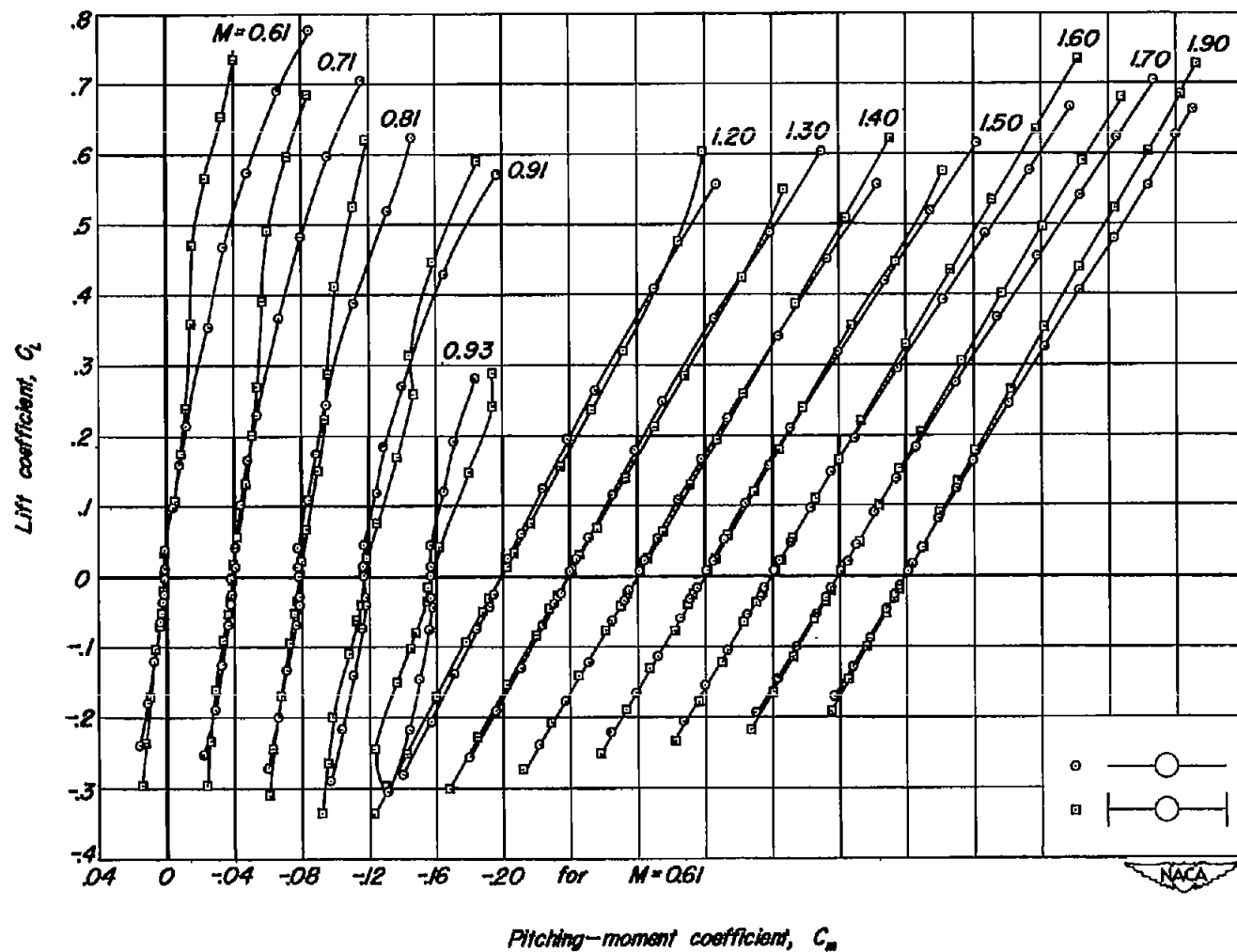
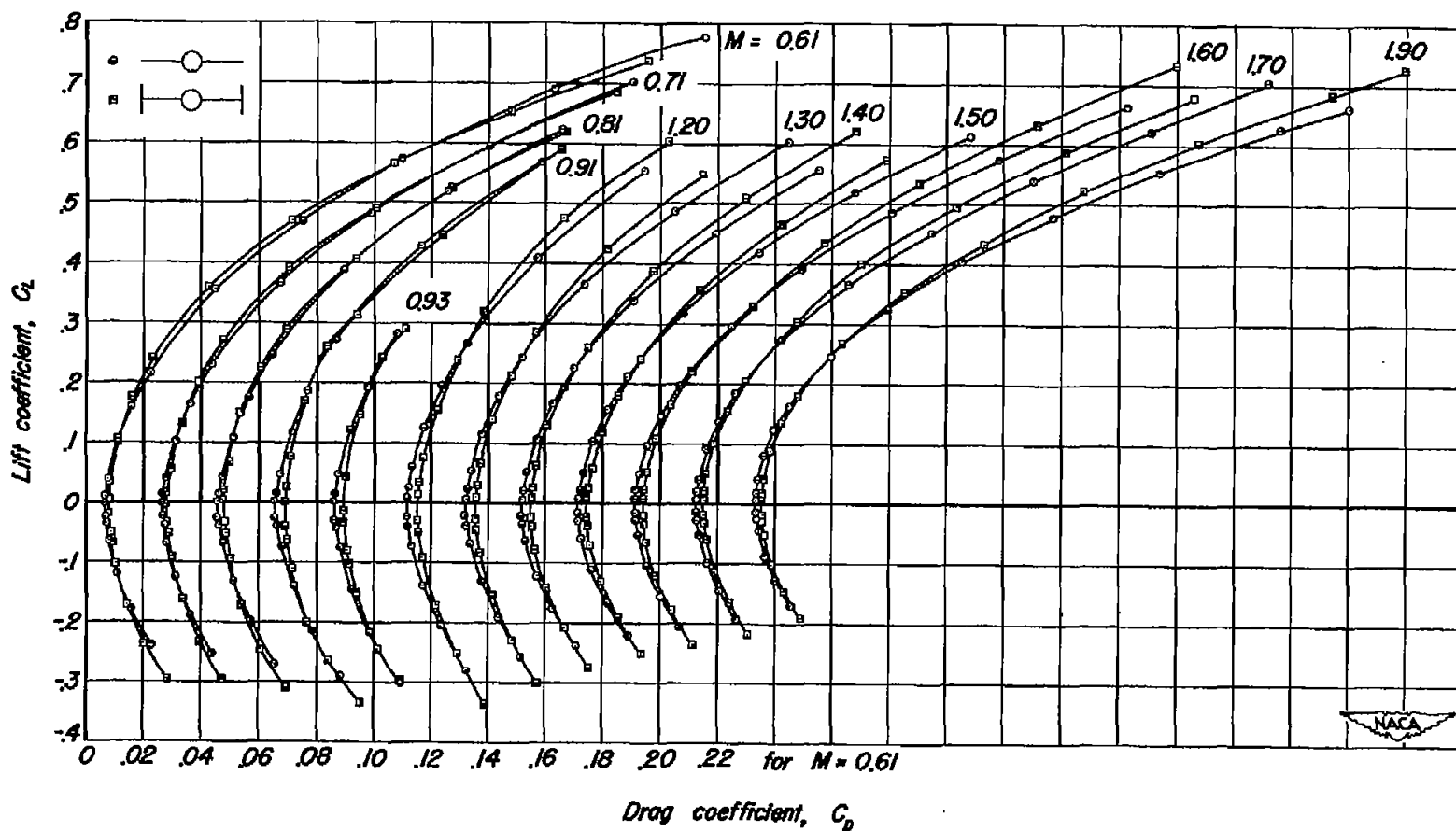
(b) C_L vs C_m

Figure 2.- Continued.



(c) C_L vs C_D

Figure 2.- Concluded.

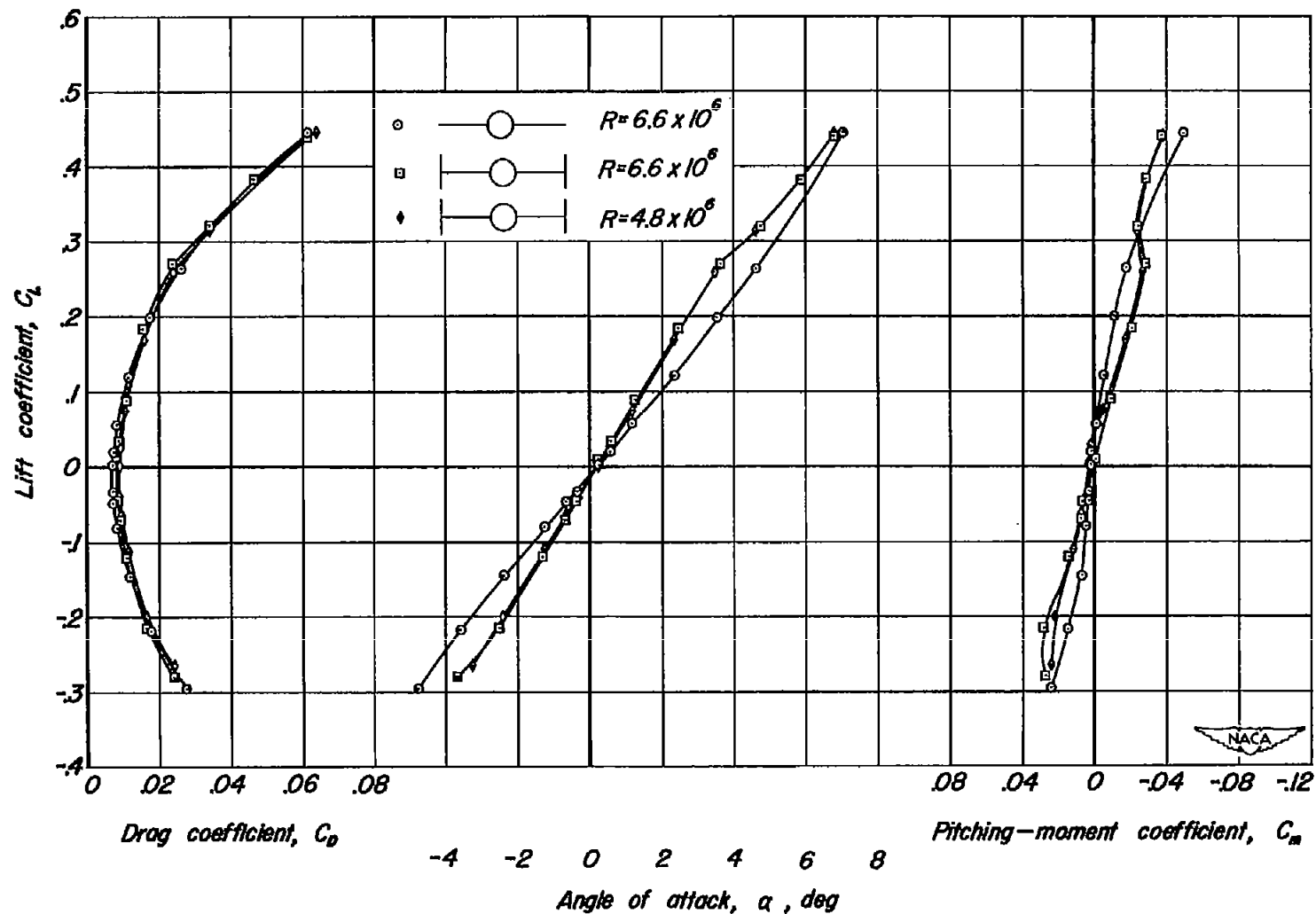


Figure 3.—The effect of Reynolds number on the variation of the aerodynamic characteristics with lift coefficient at $M=0.91$.

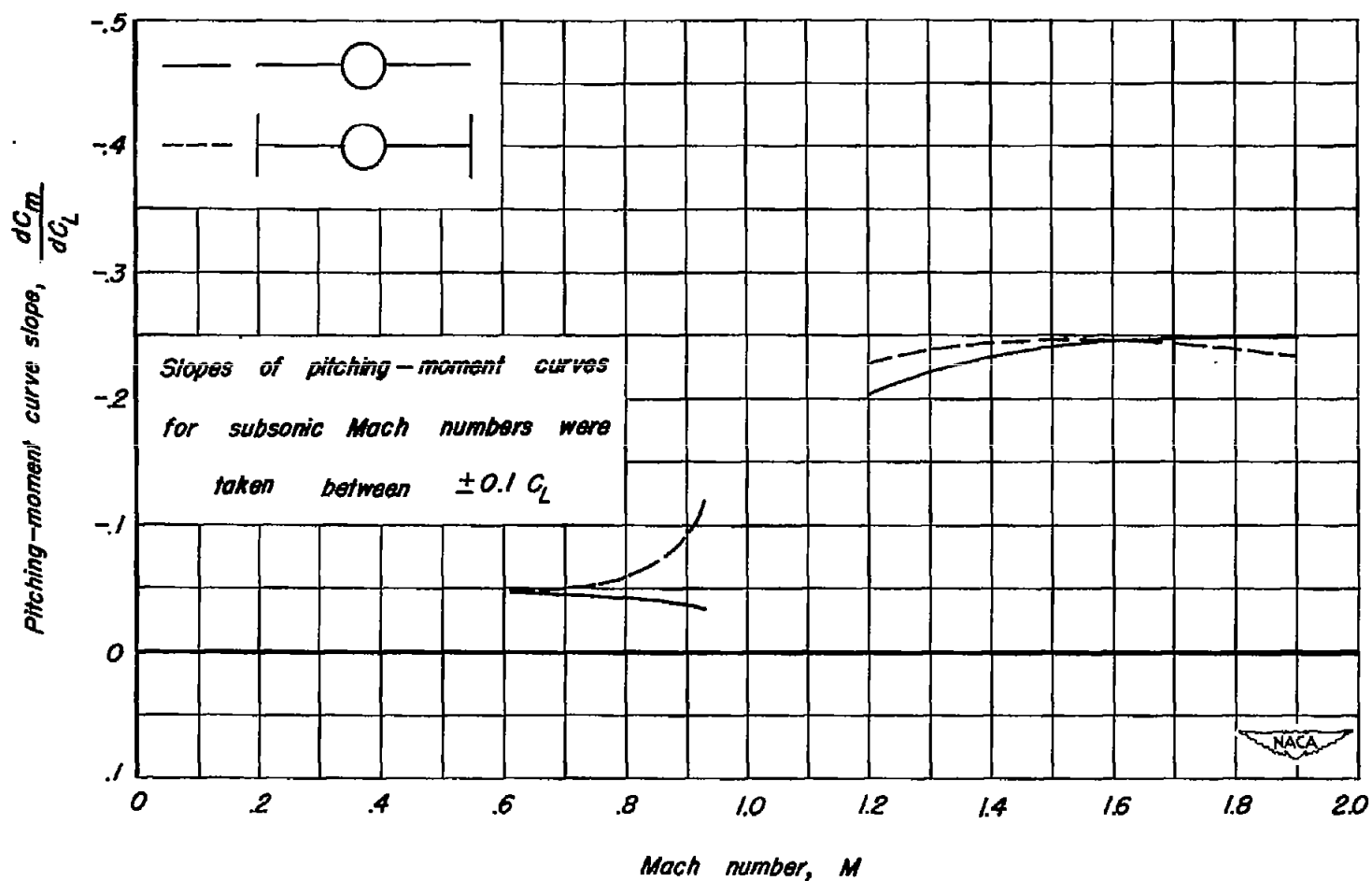


Figure 4.—The effect of end plates on the variation of static margin at various Mach numbers.

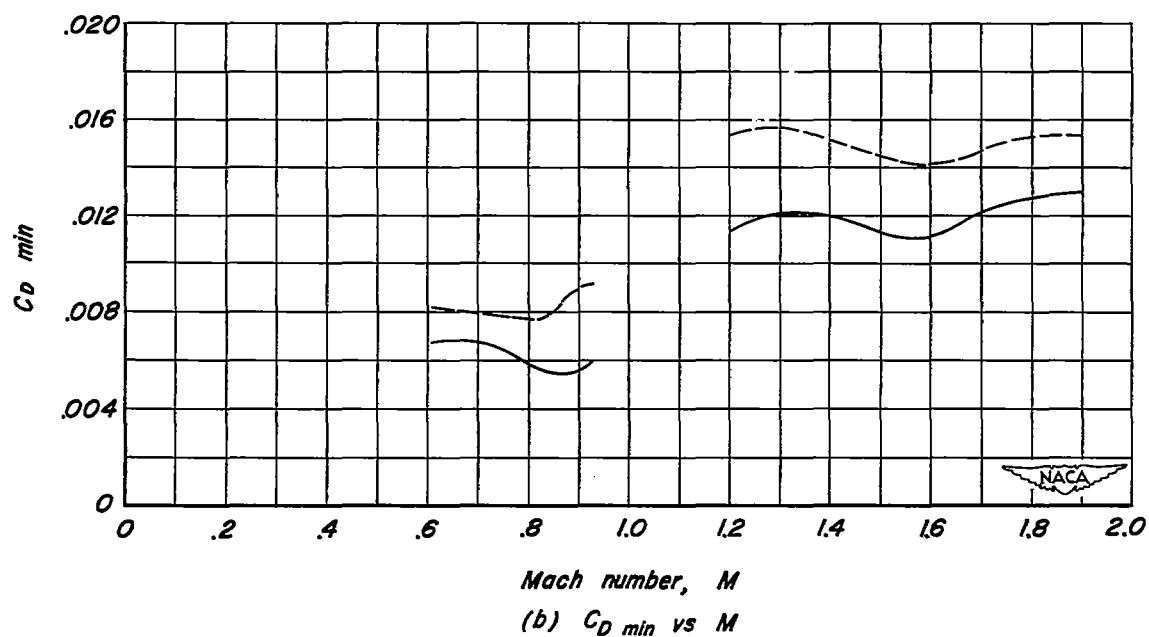
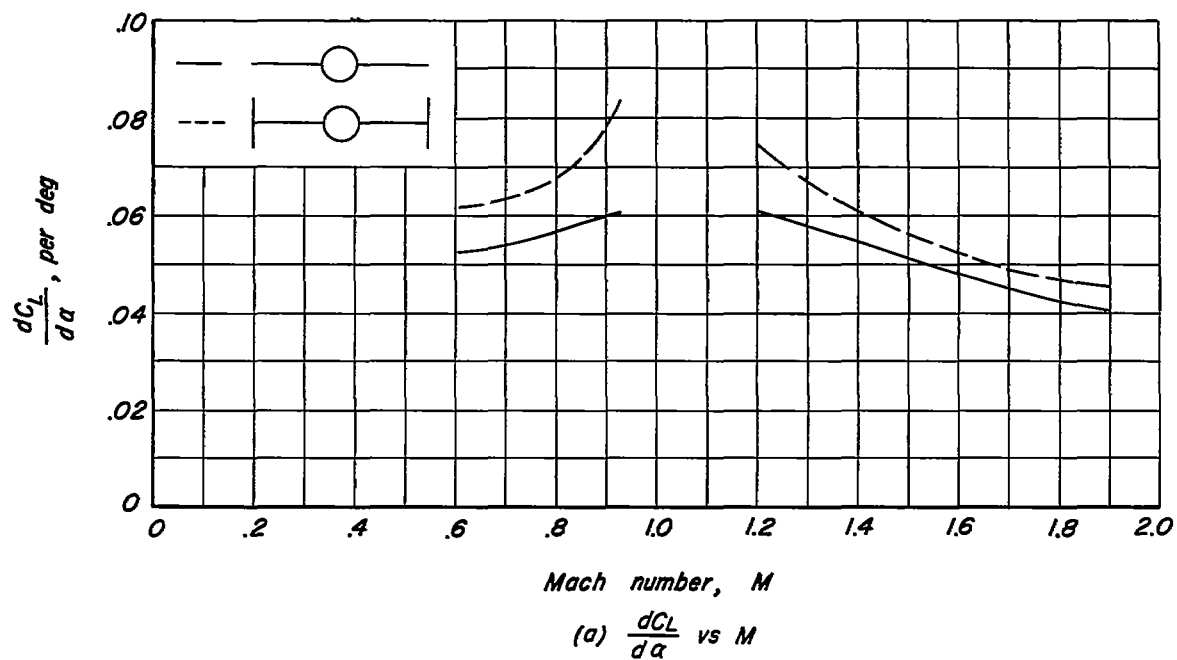


Figure 5.— The effect of the addition of end plates on various aerodynamic characteristics as a function of Mach number. Reynolds numbers as listed in table I.

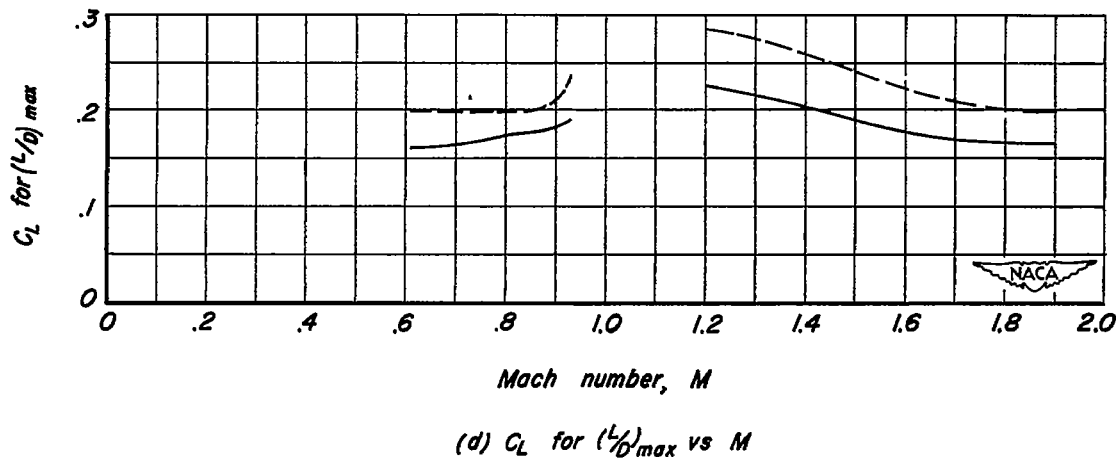
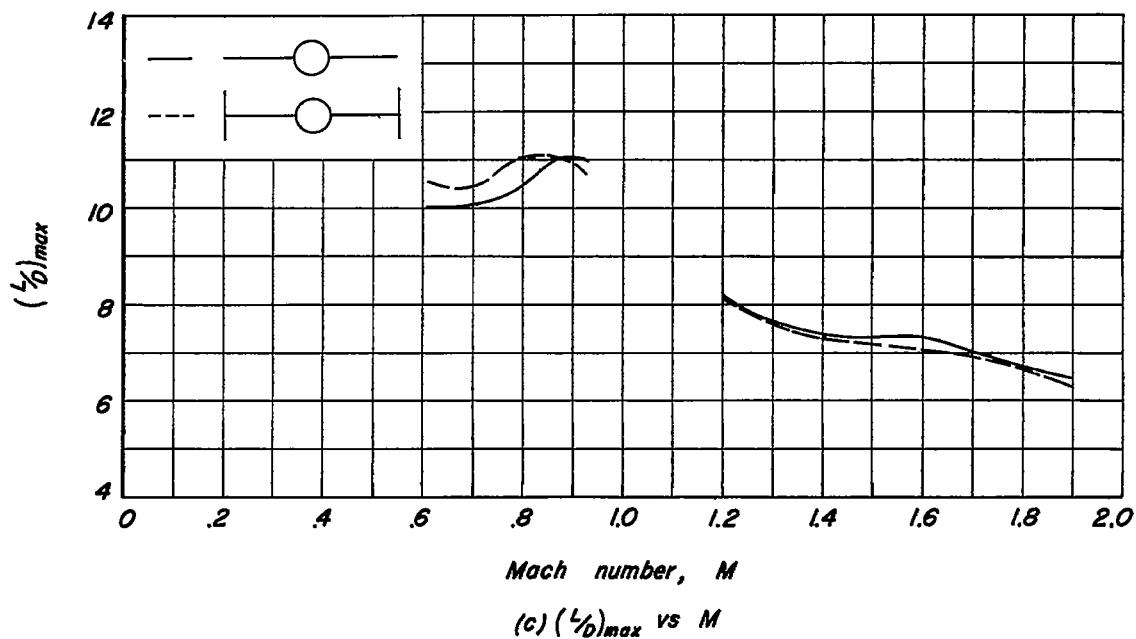


Figure 5. - Concluded.